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A REVIEW OF ELECTRON BOMBARDMENT THRUSTER SYSTEMS/SPACECRAFT FIELD AND PARTICLE INTERFACES

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ABSTRACT

This paper collates and summarizes information on the field and particle interfaces of electron bombardment ion thruster systems. Major areas discussed are the nonpropellant particles, neutral propellant, ion beam, low energy plasma, and fields. Spacecraft functions and subsystems reviewed are solar arrays, thermal control systems, optical sensors, communications, science, structures and materials, and potential control. An appendix is included to facilitate identification of specific interaction areas.

INTRODUCTION

Electron bombardment thruster (EBT) systems have been the subject of extended research and development programs. This activity has involved the normal phases of feasibility demonstration, design definition and optimization, and design verification. As the thruster system technology has matured, there has naturally occurred an increasing concern and effort associated with the integration of this technology with space systems. A major portion of the integration effort has been directed toward definition of the characteristics and constraints imposed by the particle and field effluents from EBT systems. This effort has been ongoing for over a decade-involving analytic and experimental activities by investigators at ground locations and for several space tests. These activities have provided extensive information on the field and particle efflux characteristics and their interactions with spacecraft subsystems and functions.

It is the intent of this paper to provide a convenient source of available field and particle interface information. A literature search was performed and the available data were collated. A brief summary of the characteristics, potential impact, and overall status of each major particle and field area is presented. Only published and publically available references were cited in

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the paper. Some valuable information was thereby inevitably not included. The format of the paper is intended to allow convenient updating to include other data as it becomes available.

DISCUSSION

The subject matter of the paper is organized under the major headings shown on Table 1. These headings represent the major particle and field effluents from an EBT system treated in the literature rather than any presumed priority or level of concern on the part of the author. Under each major heading, brief summaries of the characteristics and potential impact of particle and field effluents are presented. The impact is discussed with respect to the spacecraft subsystems and functions shown on Table 2.

The overall information base is cited in the appendix arranged as described on Tables 1 and 2. By reference to the appendix, the reader may rapidly identify literature pertinent to specific EBT system fields and particles and spacecraft subsystems and functions.

Nonpropellant Particle Efflux

Characteristics. - Nonpropellant particle efflux (N_P) is composed of thruster materials which are emitted due to sputtering phenomena. The operating temperatures and materials are such that negligible evaporation of thruster components occurs. Figure 1 shows the known sources of N_P. A major fraction of N_P is neutral accelerator grid material emitted due to sputtering by either low energy charge-exchange ions produced in the grid region or high energy ions from the discharge chamber. External thruster components such as the neutralizer housing and the beam shield (if used) are potential sources of trace amounts of N_P. N_P can potentially be ejected from the discharge chamber after being sputtered from the chamber walls or the positive grid.

The N_P is nearly all uncharged and is ejected from local sputter sites with spatial and energy distributions dependent upon the specific sputtering conditions and is then transported in straight line trajectories. An extremely small fraction of the N_P can potentially be converted into ions in the discharge chamber by either charge-exchange or electron impact. In this case the transport of ionic N_P is controlled by the local electromagnetic fields and the particle momentum at the time of charge-exchange reaction. One extremely important characteristic of N_P is that it is almost certain to remain on any spacecraft surface after deposition in either neutral or charged form.

Many direct measurements of N_p have been performed. Subsequent to the first measurements with a 150-cm-diameter thruster (ref. 1), data have been obtained with 5- (refs. 2 and 3), 8- (refs. 3 to 6), 15- (ref. 7), 20- (ref. 8), and 150-cm-diameter (ref. 9) mercury bombardment thrusters. Only the 15-cm thruster (ref. 7) was evaluated in space. Data were also obtained for the 12-cm cesium thruster in ground (ref. 10) and flight (ref. 11) tests. The bulk of the data were taken with two-grid thrusters, but some accelerator grid sputtering was also evaluated experimentally (ref. 4) and analytically (ref. 12) for three-grid systems where a decel electrode at zero or very low potential is placed downstream of the accelerator grid.

In all cases the preponderance of N_p was determined to be accelerator grid material, which is molybdenum for mercury bombardment thrusters and was aluminum for the 12-cm cesium thruster (ref. 11). In one case (ref. 2), N_p from the neutralizer was detected when the ion beam directly impinged on the neutralizer housing. With 8-cm thrusters a trace amount of beam shield material (titanium) was detected (ref. 5), but no carbon shield material was observed in tests with a later version thruster (ref. 6).

The N_p magnitudes vary with thruster type and operating condition. The distribution is similar, however, for all thruster types in that the net N_p deposition rate is zero on the thruster axis, goes through a maximum at an angle varying from 30° to 70°, and then very rapidly decreases with increasing angle (ref. 1). This result is due to the competing phenomena of total deposition rate of N_p and N_p removal by ion beam sputtering (both of which monotonically decrease with increasing angle).

The N_p from the accelerator grid has been modeled for two- (refs. 1, 3, and 9) and three-grid (ref. 12) accelerator systems. For these models the general form for total (not net) accelerator grid N_p is

$$U = \cos\theta \frac{R^2}{L^2} F_2 \frac{J_B}{A} \frac{f}{q} S \qquad (1)$$

where

U	total N_p arrival rate, atoms-cm^{-2}-sec^{-1}
θ	angle between normal of N_p source and direction line to deposition site
R	thruster radius, cm
L	distance between N_p source and deposition site, cm
F_2	view factor for escape of sputtered N_p from thruster

J_B ion beam current, A

A frontal area of ion beam, cm^2

f fraction of ion beam converted to charge-exchange ions

q unit charge, 1.6×10^{-19} C

S sputter yield of accelerator grid material, atoms sputtered/incident ion

Equation (1) describes the N_p from the accelerator grid, which is the major source of N_p. Deposition of N_p which has undergone charge-exchange (ref. 13) is not accounted for, although this could be done by use of the N_p-propellant charge-exchange cross sections of reference 14 and use of total charge-exchange production calculations such as presented in reference 15. This was not done herein as both extended ground (refs. 2 and 8) and flight (refs. 16 and 17) tests have not detected evidence of N_p deposition on sites where no line of sight existed to N_p sources. Additionally, no calculations of N_p from the discharge chamber or external thruster components are presented as these are felt to be negligible with respect to the accelerator grid efflux based on the low charge-exchange and ionization cross sections of most thruster materials.

Impact of N_p. - Spacecraft subsystems and functions identified in the literature as potentially impacted by N_p deposition are shown on Table 3. In all cases the potential impact was discussed in terms of the alteration of the transmittance, thermal properties, or electrical conductivities of surfaces. These physical properties are critical to many spacecraft areas. It is convenient, therefore, to discuss the impact of N_p deposition on these physical properties rather than to present a separate discussion concerning each spacecraft subsystem or function listed on Table 3.

On transmittance: Many measurements have been made of the impact of N_p deposition on transmittance. These include direct evaluation of the output of solar array segments in ground (refs. 1, 9, and 10) and flight (refs. 7, 11, and 17) tests. Data have also been taken which directly measured the total transmittance (refs. 2, 3, 9, and 10) and the transmittance as a function of wavelength (refs. 2, 3, and 10). The relationship between N_p thickness and transmittance was discussed theoretically (ref. 18). Measured values of the thickness/transmittance relationship were presented (ref. 9) for thin films of gold and aluminum and for efflux from an 8-cm mercury thruster (ref. 3). For very thin deposits ($\lesssim$2.5 nm) the N_p characteristics diverge rapidly from the bulk properties of the particular effluent. For example, at about 2.7-nm thickness the transmittance of molybdenum became wavelength specific. Reasonable agreement of theory (ref. 18) and experiment (ref. 8) was obtained on the impact of N_p on transmittance.

In summary, very small amounts of N_p deposition can strongly reduce the transmittance of surfaces. In reference 3 about a 20-percent reduction of transmittance occurred after about 2 nm of molybdenum was deposited. Unless there is line-of-sight to the known sources of neutral N_p, no spacecraft surfaces should suffer deleterious effects. For charged N_p the possibility of curved trajectories is present. In at least two space tests of EBT systems (refs. 11, 16, and 17), however, no effects of N_p on non-line-of-sight locations were detected. Reference 15 calculated the trajectories of ions in the beam of a 30-cm thruster. Even very small (fraction of an electron volt) initial downstream-directed energies prevented trajectories which led to deposition in the hemisphere upstream of the thruster grids. Sputtered material does have such directed energies, and it is probable that this is the reason no evidence of N_p deposition has been observed where no line-of-sight to effluent sources existed.

In some cases, spacecraft systems may necessarily be placed where N_p deposition could be of concern (ref. 5). In these cases the use of a beam shield appears to be an effective technique to prevent interface concerns (refs. 5 and 6).

On thermal properties: Only limited data are available on the effect of N_p deposition on the thermal properties of surfaces. The reflectance and transmittance of thin films of molybdenum and tantalum on solar cell cover plates was measured at wavelengths between 0.4 and 2.2 μm (ref. 2). The absorbtance was also calculated and presented. For the conditions of the test, strong increases in both reflectance and absorbtance of the solar cell cover were observed at all wavelengths tested, although no direct correlation with thickness was presented. The effects of thin films may be quite specific to the thermal surface upon which N_p is deposited.

Although indirect, no impact on a variety of very sensitive thermal control surfaces was noted in either the SERT II (ref. 16) or ATS-6 (ref. 11) flight tests. Reference 18 presents a theoretical discussion of the effect of N_p on thermal properties, and this analysis was used in reference 12 to evaluate the variation of solar array temperature due to very long term molybdenum deposition for a specific spacecraft configuration. Significant power losses were projected over a 10-year period due to a temperature increase caused by molybdenum deposition from an unshielded thruster with a direct line-of-sight between the array and the thruster accelerator grid.

The concerns over N_p deposition on thermal surfaces are much like those discussed previously under transmittance. Apparently, thin films of N_p can cause significant variation of thermal properties. Solutions are a matter of geometry, such as proper location of sensitive surfaces or shielding.

On surface conductivity: Change of surface electrical conductivity is of concern for science, communications systems, and electrical insulators (refs. 19 and 20) and was measured in references 1 and 9. No direct measurements of electrical surface conductivity as a function of deposition thickness were found. Reference 9, however, presented a correlation between surface conductivity and transmittance for thin layers of molybdenum from which such a correlation could be inferred; it also presented the ratio of surface to bulk resistivity as a function of deposition thickness for gold, silver, and alkali metals.

Solutions to address the impact of N_P deposition on surface conductivity should follow those previously discussed.

Neutral Propellant Efflux

Characteristics. - The major sources of neutral propellant efflux (P_0) are the ion thruster discharge chamber and the neutralizer. These particles are emitted with low energies corresponding to thruster component temperatures. P_0 is emitted from these sources with a cosine distribution. Very small amounts of P_0 emerge as high velocity neutrals produced by charge-exchange reactions in the grid region and in the ion beam. Figure 2 shows the sources of P_0, which, of course, follows straight line trajectories from the emission (or creation) source.

Direct measurement of P_0 during thruster operation is difficult due to the high energy ion beam, which tends to sputter away deposited P_0. References 6 and 10 present discussions of P_0 distributions during operation of an 8-cm mercury thruster and a 12-cm cesium thruster, respectively. An indirect evaluation (ref. 5) of the P_0 from an 8-cm thruster was made with plasma discharges off by measurement of the local pressure as a function of position.

Numerous models of the P_0 from the discharge chamber have been presented (refs. 21 to 23). In all cases the P_0 was assumed to be emitted from an extended cosine source. Reference 23 is noteworthy in that the analysis was generalized to include the effect of P_0 reemission from spacecraft surfaces to other spacecraft surfaces. Many models have also been derived (refs. 15, 21, and 24) to calculate the charge-exchange production rate in the ion beam (the high velocity neutral production rate equals, of course, the charge-exchange ion production rate). No models of P_0 from the neutralizer were found. In the opinion of the author, however, a large fraction of the propellant which enters the neutralizer is emitted as P_0.

The total P_0 produced is then the sum of that from the three sources mentioned:

$$J_{NT} = J_{ND} + J_{NN} + J_{NB} \qquad (2)$$

where

J_{NT} total P_0 produced, particles-sec^{-1}

J_{ND} P_0 emitted from discharge chamber, particles-sec^{-1}

J_{NN} P_0 emitted from neutralizer, particles-sec^{-1}

J_{NB} P_0 produced in ion beam, particles-sec^{-1}

In general, the value of J_{ND} takes the form

$$J_{ND} \approx \frac{J_B}{q}\left(\frac{1 - n}{n}\right) \qquad (3)$$

where n is the fraction of discharge chamber inlet propellant flow emitted as ions. For high accuracy the value of n should account for multiply charged ions in the beam. As previously stated the value of J_{NN} is estimated to be

$$J_N \approx \frac{\dot{m}_n}{m} \qquad (4)$$

where

m_n neutralizer inlet propellant flow, kg-sec^{-1}

m mass of propellant atom, kg

The formulae given in the literature for J_{NB} are extremely complex and the reader is referred to the literature (refs. 15, 21, and 24) for these calculations.

As discussed later, the impact of P_0 is apparently critically dependent on whether or not the P_0 remains on a spacecraft surface. Numerous analyses (refs. 21 to 23) have been performed which allow prediction of the net deposition rate of P_0 using the values of spacecraft surface temperatures, incident P_0 flux rate, and particulars of the P_0 such as vapor pressure or desorption energy. The analyses may be used to predict the conditions at which no net P_0 deposition will take place. Use of the deposition analyses in conjunction with models or direct measurements allows net surface deposition rates of P_0 to be predicted with a high degree of confidence.

Impact of P_0. - Spacecraft subsystems and functions discussed in the literature as potentially impacted by P_0 are shown on Table 3. In all but one case the literature discussed the impact of P_0 in terms of the possible alteration of spacecraft surface or material properties subsequent to P_0 deposition. Reference 13 discussed possible effects due to sunlight absorbtion and reemission into sensitive star trackers and the direct absorbtion of starlight by the P_0. The conclusions of reference 13, and additional study presented in reference 19, were that neither effect is significant and both may be justifiably ignored. Because surface and material properties are germane to many spacecraft surfaces and functions, it is again convenient to discuss the impact of P_0 in terms of particular potential physical interactions.

On chemical and metallurgical properties: A great deal of theoretical and experimental information is available concerning the chemical and metallurgical effects of mercury and cesium P_0. Hall and his coworkers have presented an extremely extensive set of theoretical analyses and experimental results concerning the interactions of mercury (refs. 18, 19, 22, 25, and 26) and cesium (refs. 27 and 28) with many typical spacecraft surfaces. The reader is referred to these publications for test conditions and detailed results. Major conclusions from these studies included: (1) no chemical or metallurgical effects were predicted or found due to mercury deposition on any organic material evaluated; and (2) of all the inorganics tested, even materials known to be extremely reactive with mercury (such as gold (ref. 26) and solder (ref. 19)) did not react at all when exposed to intense mercury neutral efflux in vacuum tests with thrusters. Although not stated in the literature, it is the author's opinion that mercury may be considered nearly totally benign to spacecraft surfaces of any type for which the conditions, previously discussed, are such that no net deposition occurs.

Some data were also presented (ref. 29) which verified the long term compatibility of liquid mercury, at various temperatures, with many candidate propellant-tank metallic and nonmetallic materials at various values of stress.

On thermal properties: Extensive analyses and data are available concerning the impact of P_0 on the thermal properties of typical spacecraft surfaces. As discussed in the previous section, no impact on thermal properties is to be expected due to chemical or metallurgical interactions. This was borne out in an extensive series of tests on thermal control surfaces, which included various black and white paints, polished aluminum, and second-surface mirrors (aluminized microsheet and quartz) (ref. 25). In these tests, carried out in-situ, no changes in emissivities or absorbtance were noted after intense exposure to mercury atom efflux. These tests were performed under conditions where no net deposition of mercury would occur. For estimates of the thermal properties in cases where net deposition is expected the reader is referred to discussions presented in references 22 and 23.

On surface conductivity and transmittance: In cases where net deposition of P_0 can occur, care must be taken to prevent undesired alterations of surface conductivity or transmittance. Mercury, for example, has a bulk resistivity 35 times greater than aluminum (ref. 19) and therefore is not desirable as a coating on antennas, electrical insulators, or dielectrics (ref. 20). An experimental correlation of mercury thickness and surface conductivity and transmittance was not found in the literature. Generally, however, the behavior of metallic P_0 should be similar to that of N_P, discussed previously, with regard to these two surface properties. Excellent discussions of these areas are presented in references 22 and 23.

Ion Beam

Characteristics. - The ion beam (P_B) mostly consists of propellant ions created in the discharge chamber plasma and accelerated to net energies dependent upon the voltages applied to the thruster and the electrons emitted from the neutralizer which provide both charge and current neutralization. For reference, this group of ions is commonly (ref. 21) referred to in the literature as thrust, or Group I, ions. A very small amount of propellant P_B is created by charge-exchange reactions in the grid region and is emitted over a large range of angles and energies dependent upon the precise location of the reaction. Traditionally (ref. 21), these ions have been called Group II ions. Figure 3 shows the sources of P_B. After emission from the thruster the ionic P_B follows nearly straight-line trajectories as the electromagnetic fields in the beam are generally too small to strongly perturb the initial trajectories. Many measurements of the spatial distribution and charge state of Group I ions and neutralizing electron temperatures and densities have been made. Recent data with the 8-cm mercury thruster were taken with and without a beam shield (refs. 5 and 6). Extensive recent measurements are also available for the mercury 30-cm thruster operated at the 3-kW power level (refs. 15 and 30) and at 6 kW (ref. 31). Group II ions, including energy distributions were measured for the 8- (ref. 5), 20- (ref. 8), and 30-cm (ref. 15) mercury thrusters.

Excellent models of Group I P_B exist. Subsequent to the early development of space-charge flow computer calculations (ref. 32) many investigators have published theoretical analyses of two- (refs. 21, 33, and 34) and three-grid (refs. 4 and 34) accelerator systems. For ease in system analysis it is convenient to use the functional forms presented in references 35 and 36, which accurately describe the Group I ions. Models of the high angle Group II ions are presented in references 15 and 21. Only a few references were found which attempted to treat the neutralizing electrons analytically (refs. 35, 37, and 38).

Impact of P_B. - Spacecraft subsystems and functions potentially impacted by P_0 are indicated on Table 3. Concerns over P_0 involve quite different physical phenomena and are discussed under the headings shown on Table 3.

On solar arrays: Solar array segments (ref. 9) and array materials (refs. 25, 26, and 39) have been subjected to intense fluxes of ion beams at energies between 1 and 3 kV with remarkably little effect noted. For example, in reference 9 a SERT II type of array segment was subjected to greater than 10^{20} ions-cm^{-2} total ion flux at greater than 3-kV energy, and no reduction in segment power output was observed. In references 25 and 26 it was reported that neither the emissivity nor absorbtance of quartz solar cell covers was affected by large total incident ion fluxes. In addition, on the ATS-6 spacecraft about one-eighth of the total solar array was in the cesium thruster ion beam (ref. 11), and no deleterious effects were noted after 92 hours of thruster operation.

On thermal properties: Extensive tests on many spacecraft materials were performed to determine the effects of ion impingement on thermal properties (refs. 25, 26, and 40). Results indicate that the emissivity of spacecraft materials is insensitive to very large (5×10^{19} ion-cm^{-2}) fluxes of ion beams. The absorbtance of most materials is also unaffected except for a few materials, such as white paints, for ion doses greater than about 10^{15} ion-cm^{-2}.

On communications: Limited information was found on the potential effects of the P_B (including the neutralizing electrons) on communications (refs. 19, 41, and 42).

In reference 41 the impact of P_B on the signal amplitude and phase of S-band signals was theoretically and experimentally evaluated for a 30-cm thruster. Changes in both properties were observed which were in good agreement with the calculations presented. Possible signal refractions were discussed in reference 41, and an estimate of this effect is presented in reference 19.

The impact of ion-electron collisions in the ion beam on uplink communications was presented in a detailed analysis in reference 42 for 30-cm mercury thruster operation. The conclusion was reached that this phenomena posed no difficulty for planetary mission communications.

It should be noted that no difficulties in uplink or downlink communications were reported in the SERT I (ref. 43), SERT II (ref. 44), or ATS-6 (ref. 11) space test flights of electron bombardment thrusters.

On science: Most of the concerns over the impact of P_B on scientific measurements are related to potential emitted electromagnetic fields. These will be discussed in a later section. As pointed out by several authors (refs. 35, 38, 45, and 46), and verified in space test (refs. 11 and 44), operation of a thruster can maintain the potential of a spacecraft near that of the local space potential. This may allow for improved measurements of low energy particles (ref. 47). Time sharing of propulsion and science data-gathering activities can remove P_B as an area of concern.

On structures and materials: No impact of P_B on spacecraft structures and materials was identified except those concerned with material removal by sputtering and special chemical effects caused by high energy propellant ion impingement on surfaces. The areas of sputtering and chemical interactions are discussed by a great many authors, and the reader is referred to the general literature for information. Reference 39 does present an extensive set of measurements and analyses of sputtering and chemical effects of P_B on many typical spacecraft materials. Again, proper location and/or shielding of structures from the beam would obviate any need for concern.

On potential control: One aspect of the impact of P_B on spacecraft potential control was mentioned above. For convenience, discussion of this area is deferred to the following section.

Low Energy Plasma Efflux

Characteristics. - The low energy plasma efflux (P_X) is produced by charge-exchange reactions between the high energy ions and neutral propellant emitted from the thruster. For convenience P_X will also be taken to include plasma produced by the neutralizer discharge. Figure 4 shows the sources of P_X. Unlike most of the particle effluxes discussed previously, the trajectories of the P_X are strongly affected by the electromagnetic fields in the ion beam and are also very sensitive to initial ion momentum.

Many measurements of the P_X have been taken with data reported for 5- (ref. 48), 8- (ref. 5), 15- (refs. 42, 49, and 50), and 30-cm mercury thrusters (refs. 15, 24, and 46) and a 12-cm cesium thruster (refs. 10 and 11). Electron and ion energy and spatial distributions and number densities have been measured. The ability of shields and electrically biased surfaces to control P_X has also been evaluated. The properties of P_X are extremely specific to thruster type and operating conditions and, therefore, the reader is referred to the cited literature for detailed experimental information. Many authors (refs. 15, 24, 46, and 48) have remarked on possible facility effects on ground test data.

Subsequent to the first analysis (ref. 21), many investigators presented models to predict the characteristics of P_X. Most authors analyzed the single-thruster case. However, the P_X from an array of thrusters was analyzed in reference 31, and this model was applied and extended in reference 51.

Calculations of the production of ionic P_X in the beam are relatively straightforward. Calculations of the production of P_X from the neutralizer were not found in the literature. The prediction of the transport of P_X is, on the other hand, very complex. Magnetohydrodynamic approaches to analyze transport have been used (refs. 31, 48, 49, and 50), and in some cases good agreement with experiment was obtained. The complexity and uncertainty of transport prediction arises from the strong impact on particle trajectories of the electromagnetic fields and initial particle momentum (refs. 15 and 24), neither of which is always available or conveniently modeled or measured.

Impact of P_X. - Table 3 shows spacecraft/P_X interactions discussed in the literature. The impact of P_X deposition and possible alteration of surface properties is probably much like that due to the deposition of previously reviewed effluxes and will not be discussed here. It should be mentioned, however, that the uncertainties in the transport make prediction of deposition difficult. The discussion will be limited to certain potential plasma interactions which could impact solar arrays, science, or spacecraft potential control.

On solar arrays: Considerable attention has been paid to potential interactions of the P_X with solar arrays (and other electrically charged spacecraft surfaces). A major impetus for this concern is the evaluation of high voltage solar arrays, which hold great promise of improved thrust system characteristics (ref. 52). Numerous calculations (refs. 49 to 51) and data from flight tests of electric thrusters (refs. 11 and 17) suggest that no significant impact on thrust system operation will occur for solar array voltages of up to several hundred volts, either positive or negative, with respect to the ion beam. Large electrically biased surfaces near the ion beam would, however, be expected to draw currents (refs. 5 and 46).

The earliest studies of high voltage arrays (refs. 53 and 54) did not have available detailed information on P_X and indicated that only very small currents would be drawn to arrays operated at voltages up to 16 kV. Further studies (refs. 48, 50, and 55), subject to the analytical difficulties previously mentioned, have indicated that significant currents can be drawn by arrays of about 500 V or more. In addition, the currents are sensitive to many specifics of arrays, such as pin holes, array-beam spacing, and cell interconnect configuration. Knowledge of high voltage array P_X interactions has been greatly extended by a series of tests with simulated solar arrays (refs. 48, 50, and 56) and tests with samples of typical solar array materials and configurations and environmental plasma conditions (refs. 55, and 57 to 60). The reader is referred to these data for specific information.

On science: A large number of science measurements have been reviewed in the literature and a detailed discussion of potential P_X interactions with all proposed instruments is beyond the scope of the paper. Excellent general reviews of many instruments and their requirements are presented in references 8, 20, 45, 47, and 61. Nearly all possible interactions of science may be eliminated by turning the thrust system off during key science measurements. Exceptions to this generality are any effects of P_X ion deposition on surfaces which are at conditions where P_X will remain after thrust system turnoff.

On potential control: It is now well documented by analysis (refs. 35, 38, and 44) are space test that full thruster operation or even operation of only the neutralizer has a strong influence over the spacecraft potential and differential charging of spacecraft surfaces. The altitude of the SERT II spacecraft (ref. 44) was too low to draw unambiguous conclusions about the impact of neutralizer-only operation. However, a series of tests with ATS-5 and ATS-6 spacecraft at geosynchronous altitude have been performed (refs. 62 to 64) and demonstrated active control of spacecraft charging-even in eclipse conditions. These tests have also indicated that a hollow cathode (ATS-6) is more effective in holding the spacecraft to near local space ground and controlling differential charging than a hot wire emitter (ATS-5).

Field Effluxes

Characteristics. - The field effluxes (F) discussed herein are the static and dynamic magnetic fields and electromagnetic fields from optical to low frequencies. Conducted F will not be discussed herein as it is very specific to power processor design.

Magnetic fields: A magnetic field is required in the discharge chamber field for efficient thruster operation, and closure of this field occurs outside of the thruster. This field represents the largest magnetic field efflux from the thrust system. Thrusters with both permanent magnets and electromagnets have been operated, and the present 8- and 30-cm mercury thruster designs are of the permanent-magnet type. The external chamber field has been measured for 5- (ref. 65), 8- (ref. 5), 20- (ref. 45), and 30-cm-diameter thrusters (ref. 65). The 20-cm and one of the 30-cm thrusters were of the electromagnet type, and data were obtained in the powered and unpowered states. As first pointed out in reference 66 and later discussed in reference 19, both quasistatic and dynamic magnetic fields could be also produced during thruster operation by asymmetric discharge currents or by the neutralizer - ion beam current loop. Data were obtained to evaluate these potential sources of magnetic field efflux in references 67 and 68 for 8- and 30-cm thrusters, respectively.

In all cases the external discharge chamber field is well represented as a magnetic dipole. For example, the 30-cm thruster field was within 5 percent of that predicted by a dipole field function for distances up to 8 meters from the thruster (ref. 65). Models to predict the overall magnetic field of an array of thrusters were presented in references 68 and 69.

Electromagnetic fields: For ease in discussion the electromagnetic fields will be arbitrarily divided into optical (those with frequencies higher than infrared) and radiofrequency (those with frequencies lower than infrared) emissions.

As mentioned previoulsy, optical reemission of sunlight is not felt to be a significant phenomenon (ref. 19). Optical emission due to the deexitation of propellant has, however, been observed. It would appear that much of the excited propellant emerges from the discharge chamber as metastables but in some cases (ref. 70) excitation was strongly affected by neutralizer electrons. Measurements were made of the magnitude of optical emissions as a function of wavelength of 8- (ref. 5), 30- (refs. 70 to 73), and 150-cm thrusters (ref. 71). In some cases the total optical power density was measured as a function of position.

The radiofrequency (rf) emissions have been measured in a few cases. Measurement with an omniantenna near a 20-cm thruster (ref. 45) detected no radiation (narrow or wide band) under steady state conditions but did find some broad band noise during the ignition of cathode discharges. Measurement of the rf from 30-cm thrusters and power processors is presented in references 74 and 75. Spectral measurements of 30-cm beam, discharge, and neutralizer plasmas were presented (ref. 76) over a range of a few kilohertz to 9 MHz. In addition, in some unpublished results from the SERT II space test of a 15-cm thruster (ref. 44), no rf was detected during thruster operation between 300 and 700 MHz or at 1700±20 MHz and 2100±20 Mhz.

Impact of F. - Table 3 identifies areas of potential impact of F discussed in the literature. It should first be pointed out that, with the exception of the static thruster magnetic field, all areas of potential impact can be obviated by the turning off of the thrust system.

The major concern about the magnetic fields is impact on magnetometer measurements (refs. 20, 45, and 74). Magnetic field cancellation strategies have been discussed in detail (refs. 20, 45, and 74). In the most recent publication (ref. 74) it was stated that with a 30-cm thruster array and a magnetometer placed at 7.5 m from the spacecraft, the magnetic field was nearly within acceptable levels. Concern over low energy plasma particle and wave detectors (ref. 45) is easily addressed by a thrust/science time sharing approach.

Optical emissions from the thrust system appear to present no difficulty. In the most recent review (ref. 72) of the many instruments reviewed, only the zodiacal photopolarimeter was considered incompatible with simultaneous thruster operation.

Likewise, rf emissions do not appear to present any problems in communication, based on the absences of observed difficulties in several space tests (refs. 11, 43, and 44) and calculation (refs. 19 and 42).

STATUS SUMMARY

The primary intent of this paper was to collate available literature on the characteristics and potential impacts of the particle and field effluxes of electron-bombardment thrust systems. The extensive and excellent information base concerning this area is evidenced in the many literature references extant. With few exceptions, sufficient information appears to exist to allow straightforward integration of the mature electron-bombardment thruster system technology with spacecraft.

For the convenience of the reader a summary, as perceived by the author, of the status of the information base and the major efflux/spacecraft interface areas is given here for each of the efflux types in Table 1.

Nonpropellant Particle Efflux

A very broad data base concerning the characteristics and potential impact of nonpropellant particle efflux exists. Models which quite accurately predict the distribution and magnitude of the efflux have been formulated and verified in both ground and flight tests.

The major concerns addressed were changes in transmittance, optical (thermal) properties, and surface conductivity which occur after N_p deposition. Such changes can be large, even with small amounts of deposited material. It seems certain, based on the data found, however, that no concerns exist for any spacecraft system not in line-of-sight from well-identified sources of N_p. To first order, the fact implies that spacecraft systems in the hemisphere upstream of the thruster accelerator grids will not suffer any deleterious effects due to N_p. This conclusion is strongly supported by considerable ground test and some rather long duration space test results. In cases where spacecraft subsystems are in the downstream hemisphere, shielding techniques have been verified which would assure no impact from N_P.

Neutral Propellant Efflux

Although difficult to measure during thruster operation, the characteristics of neutral propellant efflux are felt to be well known. Extensive effort has been applied to the generation of models of this efflux, including the effects of re-emission from spacecraft surfaces and prediction of P_0 behavior upon impact with a surface.

The major concerns over P_0 were potential chemical and metallurgical effects, impacts on thermal properties and surface conductivity, or transmittance alterations after the deposition of P_0. Very extensive theoretical and experimental results are available on these areas. It is clear that mercury P_0 is benign to nearly all spacecraft surfaces and, further is unreactive even with materials such as gold or solder under conditions when no net deposition of mercury occurs. This is nearly always the case in typical spacecraft applications. Where mercury is still of concern, such as on cold insulators, long experience with thrusters indicates simple shielding techniques are easily implemented and reliable solutions to any deposition concerns. The long term SERT II flight, which used mercury propellant, and the shorter space test on ATS-6, which used the more reactive cesium propellant appear to be totally free of any deleterious effects of P_0 deposition.

Ion Beam

The ion beam has been extensively analyzed and measured, and a strong theoretical and experimental base is available to predict the characteristics of P_B.

The major concerns about P_B evolve around material removal and chemical change due to ion impact, effects on communication, and science impact. The material property and morphology concerns can be addressed by proper placement of spacecraft systems, and the high state of knowledge of the P_B characteristics allows this to be done with low risk. The impact on communications has been analyzed and appears to be of little concern, with a question of possible communication signal refraction in the beam possibly remaining final resolution. In any case, no communication problems have been observed in several space tests. The impact on science can, of course, be obviated simply by judicious time sharing of thrust and science-taking activities. Some low energy particle physics and plasma wave analyzer experiments may be effected during thrust system operation; although many authors have pointed out that thruster, or neutralizer only, operation may allow a state of spacecraft electrostatic cleanliness which may serve to improve data quality.

Low Energy Plasma Efflux

Extensive work in modeling the low energy plasma efflux has been carried out. The production of P_X by charge-exchange reactions in the ion beam is well understood. The production of low energy ions by the neutralizer is less well known. The transport mechanisms of P_X are complex and sensitive to parameters that are difficult to estimate or measure. Much work is still in progress to better define models of P_X.

Extensive ground tests have been performed to measure the characteristics of P_X, and many authors have discussed the difficulty of obtaining totally unambiguous ground data due to possible test facility effects. In sum, the low energy P_X is at present the least understood and probably the most subject to ground experimental difficulties of any field or particle efflux from the thrust system.

The concern over P_X evolves around interactions with high voltage spacecraft surfaces, science, the effects of deposition of ionic propellant, and spacecraft potential control. Much effort has been expended to define the high voltage surface/P_X interactions. It appears at present that for solar arrays of a few hundred volts ($\lesssim$500 V) little power loss should be expected due to drain currents caused by the presence of P_X. At higher voltages the situation is presently somewhat unclear. Large charged surfaces near the thruster will draw currents, but the exact expected behavior is also unclear. The impact on science has been reviewed extensively, and the time-sharing technique mentioned previously is one technique to eliminate any concerns. It is now clear from a wealth of spacecraft data that the P_X may be used to actively control the spacecraft potential as well as strongly reduce differential charging on the spacecraft.

Field Effluxes

Magnetic and electromagnetic fields have been measured by many investigators. The discharge magnetic field is very well approximated as a dipole. Estimates have been made of the possible static magnetic fields caused by asymmetric discharges and neutralizer-beam loops, and some measurements to evaluate these fields exist. Measurements and some models also exist for optical and rf emissions from the ion beam.

The potential impact of the field emissions has been extensively studied. With the exception of the residual discharge chamber magnetic field, all impacts are eliminated when the thrust system is turned off. The discharge chamber fields of permanent-magnet thrusters would be expected to impact magnetic field measurements unless the magnetometer is placed on a boom (which may be required to avoid magnetic contamination from other spacecraft subsystems). Some impact on scientific measurements may also occur due to the discharge- or operating-caused magnetic fields, which is eliminated by a thrusting/science time-sharing philosophy.

The optical emissions appear to be of little concern as the latest measurements and analyses indicate that only one scientific instrument (zodiacal photopolarimeter) is incompatible with thrust system simultaneous operation.

The rf emissions do not appear, based on some analyses and several flight tests, to pose any problem for communication.

APPENDIX

This appendix is intended to provide easy location of information about both the characteristics and potential impacts of the field and particle effluxes shown in Table 1. These effluxes are the major headings of the tables (A-1 to A-5) under which separate sections are called out as given on Table 2. Relevant literature is cited with the date; type (experimental or theoretical); data source (ground or flight), if applicable; thruster size; and propellant type. A specific thruster size or propellant type is not indicated in many cases where the information presented was of a general nature and did not involve either data or calculation specific to thruster size or propellant.

TABLE A-1. - SPACECRAFT/THRUST SYSTEM INTERACTION AREAS

INFORMATION BASE - NONPROPELLANT PARTICLE EFFLUX

(a) Characteristics

Reference	Date	Type		Location		Thruster diameter, cm								Propellant	
		Experimental	Theoretical	Ground	Space	5	8	10	12	15	20	30	150	Hg	Cs
1	1970	•	•	•									•	•	
2	1972	•		•		•								•	
3	1973	•		•		•	•					•		•	
4	1975	•	•	•			•							•	
5	1977	•	•	•			•							•	
6	1978	•		•			•							•	
7	1970	•		•	•					•				•	
8	1973	•	•	•							•			•	
9	1971	•	•	•								•	•	•	
10	1973	•	•	•					•						•
12	1976	•	•	•			•							•	
13	1970		•											•	•
14	1977	•	•	•										•	
15	1976	•	•	•								•		•	
18	1973	•	•	•										•	•
19	1972		•											•	•
24	1975	•	•	•								•		•	
25	1970	•	•	•										•	•
26	1970	•	•	•										•	•
27	1973	•	•	•					•						•

(b) Impact evaluation

Reference	Date	Type		Location		Thruster diameter, cm								Propellant	
		Experimental	Theoretical	Ground	Space	5	8	10	12	15	20	30	150	Hg	Cs
On solar arrays															
1	1970	•	•	•									•	•	
2	1972	•		•		•								•	
3	1973	•		•		•	•					•		•	
4	1975	•	•	•			•							•	
5	1977	•	•	•			•							•	
6	1978	•		•			•							•	
7	1970	•		•	•					•				•	
8	1973	•	•	•							•			•	
9	1971	•	•	•								•	•	•	
10	1973	•		•					•						•
11	1975	•			•				•						•
12	1976	•	•	•			•							•	
13	1970		•											•	•
17	1971	•		•	•					•				•	
18	1973	•	•	•										•	•
19	1972		•											•	•
25	1975	•	•	•										•	•
26	1973	•	•	•										•	•

TABLE A-1. - Concluded.

(b) Concluded.

Reference	Date	Type		Location		Thruster diameter, cm								Propellant	
		Experimental	Theoretical	Ground	Space	5	8	10	12	15	20	30	150	Hg	Cs
On thermal control															
2	1972	●		●		●								●	
6	1978	●		●			●							●	
8	1973	●	●	●							●			●	
11	1975	●			●				●						●
16	1971	●		●	●					●				●	
18	1973		●											●	●
19	1972		●												
20	1973		●												
On optical sensors															
1	1970	●	●	●									●	●	
2	1972	●		●		●								●	
3	1973	●		●		●	●					●		●	
4	1975	●	●	●			●							●	
5	1977	●	●	●			●							●	
6	1978	●		●			●							●	
7	1970	●		●	●					●				●	
8	1973	●	●	●							●			●	
9	1971	●	●	●								●	●	●	
10	1973	●		●					●						●
11	1975	●			●				●						●
12	1976	●	●	●			●							●	
13	1970		●												
18	1973	●	●	●										●	●
19	1972		●												
25	1970	●	●	●										●	●
26	1973	●	●	●										●	●
On communications															
1	1970	●		●									●	●	
9	1971	●		●								●	●	●	
19	1972		●												
20	1973		●												
On science															
1	1970	●		●									●	●	
9	1971	●		●								●	●	●	
18	1973	●	●											●	●
19	1972		●												
20	1973		●												
45	1973		●												

TABLE A-2. - SPACECRAFT/THRUST SYSTEM INTERACTION AREAS

INFORMATION BASE - NEUTRAL PROPELLANT EFFLUX

(a) Characteristics

Reference	Date	Type		Location		Thruster diameter, cm								Propellant	
		Experimental	Theoretical	Ground	Space	5	8	10	12	15	20	30	150	Hg	Cs
5	1977	●		●			●							●	
6	1978	●		●			●							●	
10	1973	●		●					●						●
15	1976	●	●	●								●		●	
18	1973	●	●	●										●	●
19	1972		●												
21	1968		●												
22	1969		●											●	●
23	1969		●												
24	1975	●	●	●								●		●	

(b) Impact evaluation

Reference	Date	Type		Location		Thruster diameter, cm								Propellant	
		Experimental	Theoretical	Ground	Space	5	8	10	12	15	20	30	150	Hg	Cs
On solar arrays															
11	1975	●			●				●						●
17	1971	●			●					●				●	
18	1973	●	●	●										●	●
22	1969		●											●	●
23	1969		●											●	
25	1970	●	●	●										●	●
26	1970	●	●	●										●	●
27	1973	●	●	●											●
28	1973	●	●	●											●
On thermal control															
16	1971	●			●					●				●	
18	1973	●	●	●										●	●
19	1972		●											●	●
20	1973		●											●	
22	1969		●											●	●
23	1969		●											●	
25	1970	●	●	●										●	●
26	1970	●	●	●										●	
27	1970	●	●	●											●
28	1973	●	●	●											●

TABLE A-2. - Concluded.

(b) Concluded.

Reference	Date	Type		Location		Thruster diameter, cm								Propellant	
		Experimental	Theoretical	Ground	Space	5	8	10	12	15	20	30	150	Hg	Cs
On optical sensors															
6	1978	•		•			•							•	
13	1970		•											•	•
19	1972		•											•	•
22	1969		•											•	•
27	1973	•	•	•					•						•
28	1973	•	•	•											•
On communications															
19	1972		•											•	•
20	1973		•											•	
22	1969		•											•	•
On science															
13	1970		•											•	•
19	1972		•											•	•
20	1973		•											•	
On structures and materials															
18	1973	•	•											•	•
19	1972		•											•	•
22	1969		•											•	•
25	1970	•	•	•										•	•
26	1970	•	•	•										•	•
27	1973	•	•	•											•
28	1973	•		•											•
29	1973	•		•										•	

TABLE A-3. - SPACECRAFT/THRUST SYSTEM INTERACTION AREAS

INFORMATION BASE - ION BEAM

(a) Characteristics

Reference	Date	Type		Location		Thruster diameter, cm								Propellant	
		Experimental	Theoretical	Ground	Space	5	8	10	12	15	20	30	150	Hg	Cs
4	1975	•	•	•			•							•	
5	1977	•	•	•			•							•	
6	1978	•		•			•							•	
8	1973	•	•	•							•			•	
15	1976	•	•	•								•		•	
21	1968		•											•	
30	1974	•	•	•								•		•	
31	1977	•	•	•								•		•	
32	1966		•											•	
33	1968		•											•	
34	1974		•											•	
35	1970	•	•	•											
36	1971		•												
37	1974		•												

(b) Impact evaluation[a]

Reference	Date	Type		Location		Thruster diameter, cm								Propellant	
		Experimental	Theoretical	Ground	Space	5	8	10	12	15	20	30	150	Hg	Cs
On solar arrays															
9	1971	•		•								•	•	•	
11	1975	•			•				•						•
25	1970	•	•	•										•	•
26	1970	•	•	•										•	•
39	1972	•		•										•	
On thermal control															
25	1970	•	•	•										•	•
26	1970	•	•	•										•	•
40	1972	•		•										•	
On communications															
11	1975	•			•				•						•
19	1972		•												
41	1976	•	•	•								•		•	
42	1972	•	•									•		•	
43	1965	•			•			•						•	
44	1970	•			•					•				•	
On science															
11	1975	•			•				•						•
35	1970	•	•	•											
38	1969	•	•	•											
44	1970	•			•					•				•	
45	1973	•	•	•							•			•	
47	1975		•												
On structures and materials															
39	1972	•	•	•										•	

[a]Evaluation of impact on potential control is given in table A-4.

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TABLE A-4. - SPACECRAFT/THRUST SYSTEM INTERACTION AREAS

INFORMATION BASE - LOW ENERGY PLASMA EFFLUX

(a) Characteristics

Reference	Date	Type		Location		Thruster diameter, cm								Propellant	
		Experimental	Theoretical	Ground	Space	5	8	10	12	15	20	30	150	Hg	Cs
5	1977	•		•			•							•	
10	1973	•		•					•						•
11	1975	•			•				•						•
15	1976	•	•	•								•		•	
21	1968		•												
24	1975	•	•	•								•		•	
31	1977	•	•	•								•		•	
44	1970	•		•	•					•				•	
46	1975	•	•	•								•		•	
48	1977	•	•	•		•								•	
49	1975	•	•	•						•				•	
50	1976	•	•	•						•				•	
51	1977		•									•		•	

(b) Impact evaluation[a]

Reference	Date	Type		Location		Thruster diameter, cm								Propellant	
		Experimental	Theoretical	Ground	Space	5	8	10	12	15	20	30	150	Hg	Cs
On solar arrays															
5	1977	•		•			•							•	
11	1975	•			•				•						•
17	1971	•			•					•				•	
48	1977	•	•	•		•								•	
49	1975	•	•	•						•				•	
50	1976	•	•	•						•				•	
51	1977		•									•		•	
53	1970		•												
54	1970		•												
55	1974	•	•	•											
56	1969	•	•	•											
57	1975	•	•	•											
58	1974	•	•	•											
59	1972	•	•	•											
60	1972	•	•	•											
On science															
8	1973	•	•	•							•			•	
20	1973		•												
45	1973	•	•	•							•			•	
47	1975	•	•	•											
61	1975		•												
On potential control															
35	1970	•	•	•											
38	1969	•	•	•											
44	1970	•			•					•				•	
46	1975	•	•	•								•		•	
62	1977	•	•		•				•						•
63	1977	•	•		•				•						•
64	1978	•	•		•				•						•

[a]Evaluation of impact on thermal control is given in table A-2.

TABLE A-5. - SPACECRAFT/THRUST SYSTEM INTERACTION AREAS

INFORMATION BASE - FIELD EFFLUX

(a) Characteristics

Reference	Date	Type		Location		Thruster diameter, cm								Propellant	
		Experimental	Theoretical	Ground	Space	5	8	10	12	15	20	30	150	Hg	Cs
5	1977	•		•			•							•	
19	1972		•												
45	1973	•	•	•							•			•	
65	1973	•		•								•		•	
66	1970	•	•	•											
67	1978	•		•			•							•	
68	1975	•	•	•								•		•	
69	1972		•									•		•	
70	1971	•		•								•		•	
71	1971	•		•								•	•	•	
72	1975	•	•	•								•		•	
73	1972	•		•								•		•	
74	1973	•		•								•		•	
75	1973	•		•								•		•	
76	1973	•		•								•		•	

(b) Impact evaluation

Reference	Date	Type		Location		Thruster diameter, cm								Propellant	
		Experimental	Theoretical	Ground	Space	5	8	10	12	15	20	30	150	Hg	Cs
On optical sensors															
19	1972		•												
70	1971	•		•								•		•	
71	1971	•		•								•	•	•	
72	1975	•	•	•								•		•	
73	1972	•		•								•		•	
On communications															
11	1975	•			•				•						•
19	1972		•												
42	1972		•									•		•	
43	1968	•			•			•						•	
44	1970	•			•					•				•	
On science															
5	1977	•		•			•							•	
19	1972		•												
20	1973		•												
45	1973	•	•	•							•			•	
70	1971	•		•								•		•	
72	1975	•	•	•								•		•	
74	1973	•		•								•		•	
75	1973	•		•								•		•	

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TABLE 1. - ELECTRON BOMBARDMENT THRUSTER SYSTEM PARTICLE AND FIELD EFFLUXES

Nonpropellant particles, N_P
Neutral propellant, P_0
Ion beam, P_B
Low energy plasma, P_X
Fields, F

TABLE 2. - SPACECRAFT SUBSYSTEMS AND FUNCTIONS

Solar arrays
Thermal control
Optical sensors
Communications
Science
Structures and materials
Potential control

TABLE 3. - POTENTIAL INTERACTION AREAS PRESENTED IN LITERATURE

Spacecraft subsystem or function	EBT system particle and field efflux				
	Nonpropellant particles, N_P	Neutral propellant, P_0	Ion beam, P_B	Low energy plasma, P_X	Fields, F
Solar arrays	•	•	•	•	
Thermal control	•	•	•	•	
Optical sensors	•	•	•	•	•
Communications	•	•	•	•	•
Science	•	•	•	•	•
Structures and materials		•	•	•	
Potential control			•	•	

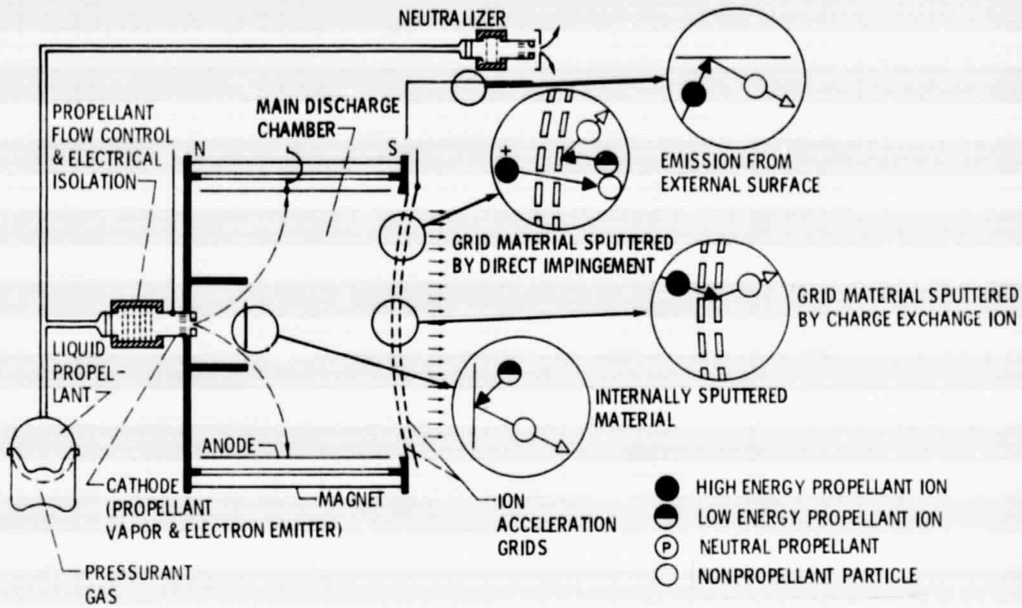

Figure 1. - Sources nonpropellant particle efflux N_P.

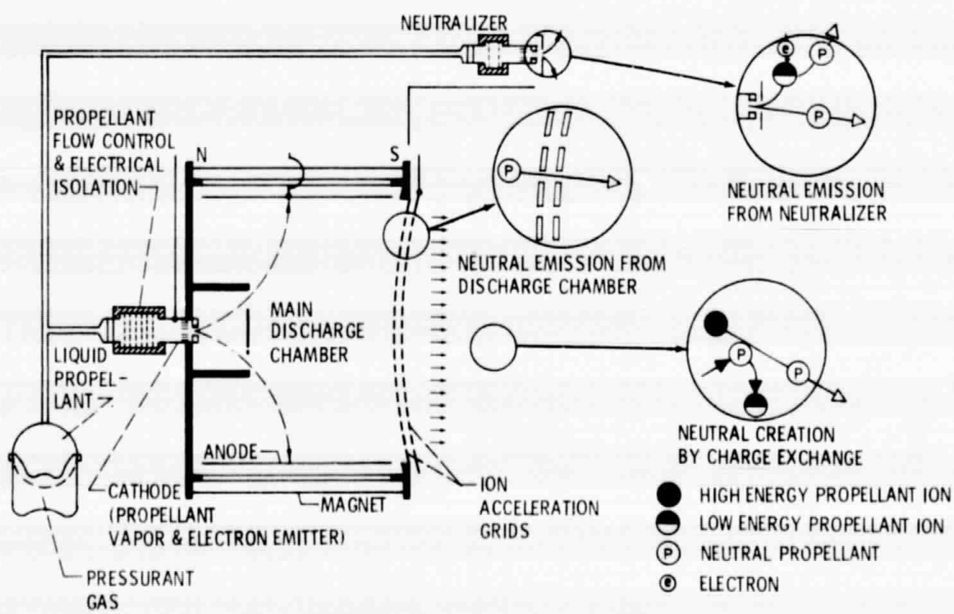

Figure 2. - Sources of neutral propellant efflux P_0.

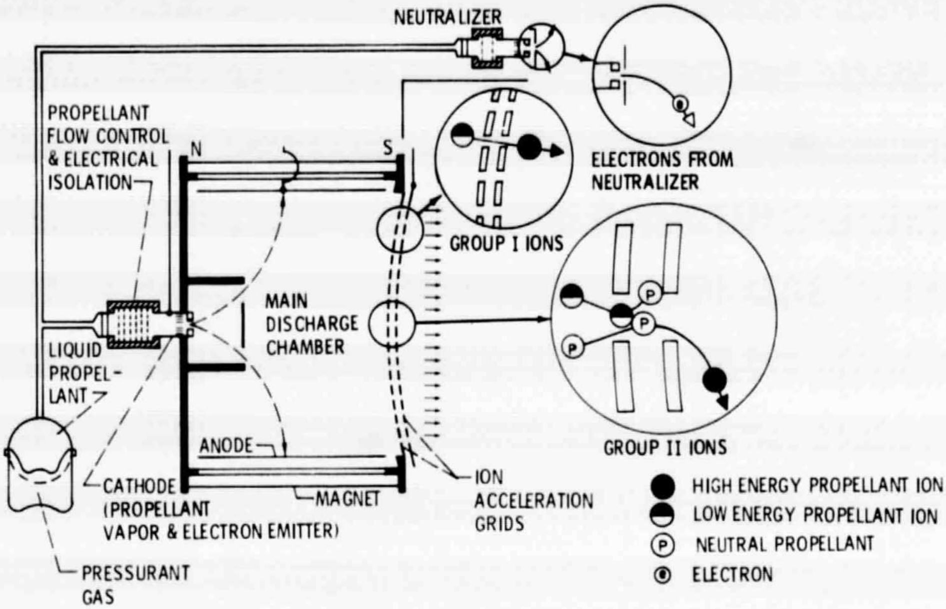

Figure 3. - Sources of ion beam efflux P_B.

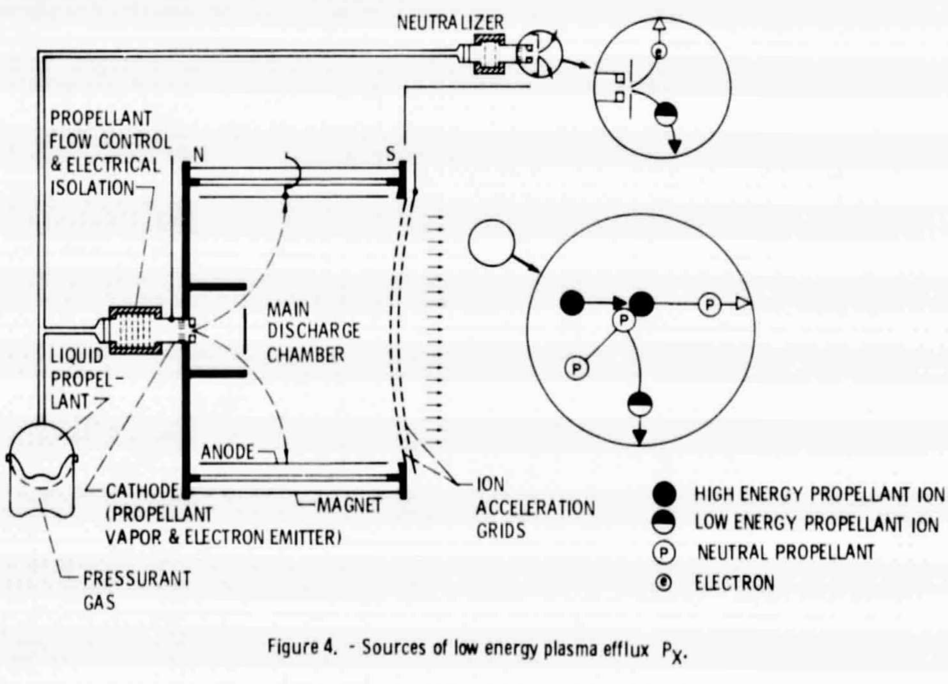

Figure 4. - Sources of low energy plasma efflux P_X.